

Black Powder

model rocket engines

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(Technical Editor's Note: The following article by Trip Barber is a very technical discussion of the model rocket engines we all use. Read it, even though it gets fairly heavy in most places (You can just skip over the equations.): there's a lot of good information about engine characteristics in it that may aid you in competition. [Please note, though, that this is not intended to be a do-it-yourself article on making rocket engines.]

Black powder (gunpowder) is the oldest rocket fuel known to man, being used for this purpose in Europe as early as the eighth century. Since World War I, however, the development of higher-energy castable double-base and composite propellants has made it obsolete for professional rocket applications, except in igniter squibs. Its low cost, smooth burning characteristics at low pressures, low exhaust product toxicity, and mechanical properties all make it the safest and cheapest choice for model rocket engines, but because of its low energy content little attention has been paid to describing its properties as a rocket fuel in the professional literature. This article outlines the theory of solid propellant rockets with particular emphasis on the characteristics of current black powder model rocket engines. All of the characteristics given in this article are based on experimental measurements performed

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over the last 5 years by Ed Brown of Estes Ind., Scott Layne (NAR 7901), Mark Stutman (NAR 18569), and myself, or on data researched in the slim available literature. The data do not apply to the Enerjet or Thunderbolt engines, which do not use black powder. Metric units are used throughout. A list of equation variables and their units is presented at the end.

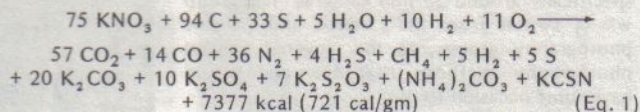
A normal model rocket engine is made up of six basic parts (Figure 1): casing, nozzle, propellant grain, delay charge, ejection charge, and end cap. The casing is generally a convolutely-wound paper tube with a rupture pressure several times that normally developed in the engine. The nozzle is made of hard clay densely packed to a precise size. The propellant grain is a single body made up of all the engine's highly compressed black powder fuel. These three are all a booster-type engine contains. In other engines, on the end of the grain away from the nozzle is a precisely-measured length of dense and slow-burning chemical used to delay the ejection charge ignition. The ejection charge is a loosely-packed, fast-burning chemical which is held in place by the clay or paper end cap.

A solid propellant represents stored energy which can be made available through exothermic chemical reaction (combustion) to produce the gas kinetic energy used by rockets for propulsion. It must be physically and chemically stable at ambient temperatures and react only when enough energy is applied to its surface to raise it above ignition temperature, which is about 500°K for black powder. Other desirable properties include high density (to minimize casing volume and weight), resistance to change in ambient conditions, and good storability. At 1.91 g/cc, the density of black powder is quite high, but, as discussed later, it is sensitive to storage conditions, tending to absorb moisture from the air with unknown subsequent effects on performance and to vary in performance somewhat with changes in ambient temperature.

Black powder is a composite-type propellant, made by mechanically mixing fuel (charcoal and sulfur) and oxidizer (potassium nitrate). Like all solid propellants, it does not use oxygen from the air for combustion. The powder most commonly used in model rocket engines since 1972 is made by CIL, a Canadian company, with the following composition by weight:

74.0% KNO₃
10.4% S
15.6% charcoal (70% C, 5% H₂O, 5% H₂, 17% O₂, and 3% ash)

This fuel is believed to burn with the following approximate chemical reaction:



This produces exhaust products that at room temperature are 40% gaseous and 60% solid (by weight), but at the combustion chamber temperature (T_c) of 1900°K, which is typical of all black powder engines, all but K₂SO₄ are gaseous. This T_c value was obtained by direct measurement with a platinum-rhodium thermocouple. Some standard thermochemical calculations give the properties of the exhaust products to be approximately as follows: M = 63.5; R = 13.348 m°/K; c = 9.2 cal/gm °K; and γ = 1.276. These properties are important in the equations for predicting engine and nozzle performance.

The basic equation for a rocket engine is Newton's Third Law, in a form which states that the engine thrust (or the change of the rocket's momentum) is equal and opposite to the change in momentum of the exhaust gases across the nozzle; i.e., that for every action there is an equal and opposite reaction.

$$F = \frac{\Delta(mv)}{\Delta t} = \frac{\Delta m}{\Delta t} v = \dot{m}_{\text{ex}} v_{\text{ex}} \quad (\text{Eq. 2})$$

The nozzle changes the 721 cal/gm of energy released by the propellant combustion and stored in the gaseous products as thermal energy into kinetic energy, or velocity and momentum. Some successful work has been done on augmenting thrust through the $\dot{m}_{\text{ex}}$ term by injecting freon, water, etc. into the exhaust or by using tandem engines in which casing mass is eroded from the lower engine. Under certain conditions, the increase in $\dot{m}_{\text{ex}}$ more than offsets the accompanying decrease in v_{ex} .

A standard (deLaval-type) model rocket nozzle has 3 parts (Figure 2): a converging section, where the flow is accelerated from zero to sonic speed; a throat, where the flow velocity is equal to the speed of

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EQUATION VARIABLES AND CONSTANTS

A	— m ²	area
c	— cal/gm. °K	specific heat at constant pressure
CF	—	nozzle thrust coefficient
d	— m	diameter
e	—	base of natural logarithms (2.71828)
F	— kg-m/sec ² (N)	thrust
g	— m/sec ²	gravitational constant (9.806)
I _{sp}	— sec	specific impulse
I _T	— Nt-sec.	total impulse
Kn	—	burning area ratio (A _b /A _t)
m	— kg	mass
ṁ	— kg/sec.	mass flow rate
M	—	molecular weight
p	— atm.	pressure (1 atm = 1.013 x 10 ⁴ Nt/m ²)
r	— m/sec	burning rate
R	— m/°K	specific gas constant
t	— sec	time
T	— °K	temperature
v	— m/sec.	velocity
α	— deg.	nozzle divergence half-angle
γ	—	ratio of specific heats
e	—	nozzle area ratio (A _{ex} /A _t)
ρ	— kg/m ³	density

VARIABLE SUBSCRIPTS

amb	— local atmospheric
b	— burning surface
c	— combustion chamber
ex	— nozzle exit
i	— pre-firing
p	— propellant
t	— nozzle throat
x	— position along nozzle axis

sound; and a diverging section, where the flow is accelerated to supersonic speeds. As shown in Figure 2, while the area does strange things, because of peculiarities in gas dynamics at supersonic speeds, there is a smooth exchange of stored energy (temperature and pressure) for kinetic energy (gas velocity) along the nozzle length. If we make the simplifying but not completely accurate assumption that the engine's exhaust products are a perfect gas (ignoring the solid particles), then these equations govern flow throughout the nozzle:

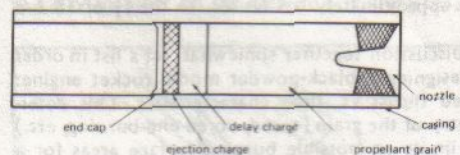


Figure 1. Typical cored-end-burning non-booster model rocket engine.

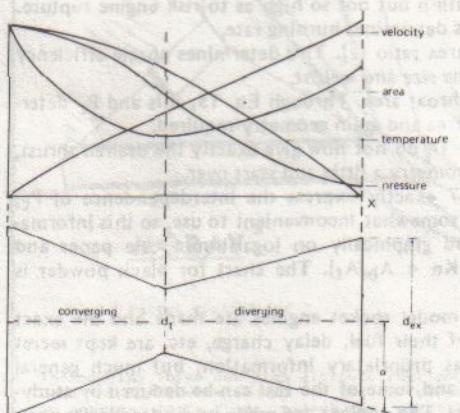


Figure 2. Typical nozzle with exhaust gas characteristics.

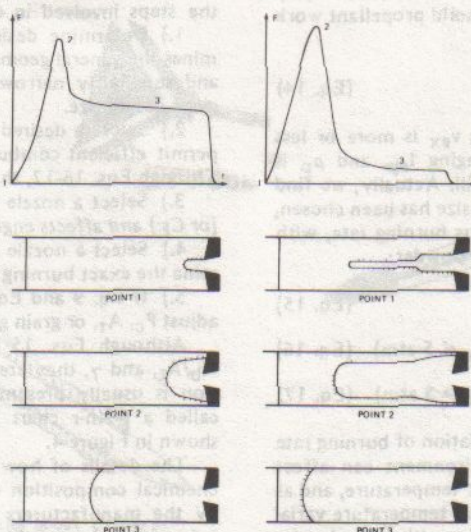


Figure 3. Thrust level versus burning surface development for typical cored-end-burning (left) and cored (right) engines.

$$P_x = \rho_x R T_x \quad (\text{perfect gas law}) \quad (\text{Eq. 3})$$

$$\rho_{x1} A_{x1} v_{x1} = \rho_{x2} A_{x2} v_{x2} \quad (\text{conservation of mass}) \quad (\text{Eq. 4})$$

$$T_c/T_x = (P_c/P_x)^{(\gamma-1)/\gamma} \quad (\text{isentropic flow}) \quad (\text{Eq. 5})$$

While at the nozzle throat

$$v_t = \sqrt{\frac{2g\gamma}{\gamma+1} R T_c} \quad (\text{Eq. 6})$$

and at any point downstream of the throat

$$v_x = \sqrt{\frac{2g\gamma}{\gamma-1} R T_c \left[1 - \left(\frac{P_x}{P_c} \right)^{(\gamma-1)/\gamma} \right]} \quad (\text{Eq. 7})$$

The length of the nozzle converging section and its steepness are not important (on black powder engines this section is virtually non-existent), but the diverging section must be smooth and gradual to maintain efficient flow ($\alpha = 10^\circ$ usually), and the longer it is, up to the point where it produces an exit pressure equal to atmospheric, the more completely and efficiently it can convert the gas energy to velocity. Unfortunately, long nozzles mean high nozzle and casing weights, so model rocket engines usually compromise and use a slightly underexpanded nozzle, with $P_{ex} > P_{atm}$. Estes and Centuri engines use an area ratio $\epsilon = 2$.

Rocket engines will work if the combustion gases are at any pressure greater than the surroundings, but they are only efficient when the chamber pressure is sufficient to produce supersonic (choked) flow, or (for black powder) P_c at least 1.9 times local atmospheric pressure. A simple empirical formula for working back from the measured thrust of a black powder engine to get its chamber pressure is

$$P_c = (.00001113F / d_t^2)^{.978} \quad (\text{Eq. 8})$$

Some typical values of d_t for Estes engines are: 1/4A3-, 1/2A3-, A3-, A8-, B6-, and C6-.002845 m; B4-.003302 m; B14-.003556 m; and D12-.004191 m. This formula only works for $P_c \geq 5$ atm.

There are many ways to express the basic action-reaction concept of rocket thrust, and a second common one is

$$F = C_F P_c A_t \quad (\text{Eq. 9})$$

Here P_c must be in units of Nt/m² (1 atm = 1.013 x 10⁴ Nt/m²). This formula says that thrust is equal and opposite to the algebraic sum of all of the exhaust gas-pressure forces acting on every surface of the engine's nozzle and combustion chamber; i.e., thrust is the result of un-

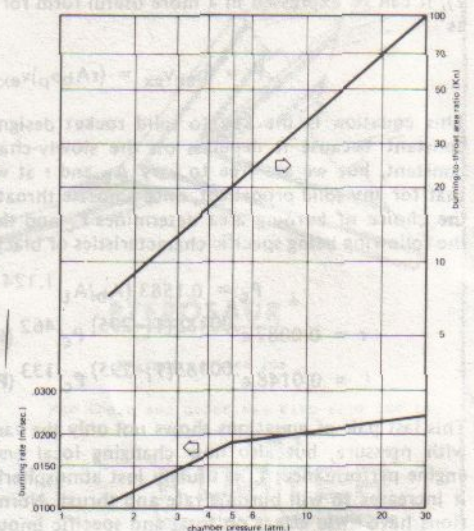


Figure 4. p-Kn-r chart for black powder at 295° K.

balanced pressure forces. The sum of pressures is equal to $C_F P_c A_t$, the pressure force across the throat times a coefficient which depends on the efficiency of the nozzle and of the gas flow through it. This coefficient is different for each thrust level of each engine and can be calculated from the nozzle and gas characteristics. This shows that engine thrust will increase with chamber pressure and with nozzle throat area; however, for a given engine, pressure decreases as nozzle area increases (the gas has an easier time escaping) so there must be a tradeoff. Because of their method of manufacture, model rocket engines cannot tolerate high chamber pressures, a fact which reduces their potential performance; like most solid fuels, black powder burns more efficiently at higher pressures, as Eqs. 18-19 show. In addition, the fact that these engines burn unevenly, with many large particles being ejected in the exhaust, means that the nozzle throat must be kept large enough that it will not become clogged.

In discussing the power produced by a rocket engine or fuel, the terms "total impulse" and "specific impulse", respectively, are used. Total impulse (I_T) simply expresses the amount of energy imparted to the rocket by the thrust of its engine, and is equal to the engine's average thrust multiplied by its burn time, or more exactly

$$I_T = \int_0^t F dt \quad (\text{Eq. 10})$$

This is equal to the area under the graph of thrust versus time. Its value as measured by the NAR Standards and Testing Committee is used to divide model rocket engines into the familiar power classes. Current S&T rules require that the manufacturers produce engines with I_T far enough below the maximum for their class that no more than 3% will ever have more impulse than this maximum due to manufacturing variations; i.e. the engines must have an average I_T two standard deviations less than the maximum. To meet this standard, most engines actually deliver about 10% less impulse than their manufacturers' catalog values; the typical 10 Nt-sec. C engine delivers an average of 9.1 Nt-sec.

Specific impulse (I_{sp}) is a means of expressing the potential energy of a rocket fuel. It is simply the amount of total impulse that would be delivered by burning one unit of weight (not mass) of the propellant.

$$I_{sp} = I_T / m_p g = I_T / w_p \quad (\text{Eq. 11})$$

Modern high-energy solid fuels deliver I_{sp} of 300 sec. or so, while black powder averages from 80 sec. down in most engines. Enerjet engines deliver around 200 sec.; this is why they can deliver so much more power per unit of engine weight. The direct relationship between I_{sp} and rocket energy is obvious from the equations:

$$v_{ex} = I_{sp} g \quad (\text{Eq. 12})$$

$$F = \dot{m}_{ex} v_{ex} = \dot{m}_{ex} I_{sp} g \quad (\text{Eq. 13})$$

These are only exact for an ideally-expanded nozzle (exit pressure equal to atmospheric), but they are close enough for most model rocket engines.

Returning now to the basic momentum-transfer thrust equation (Eq. 2), it can be expressed in a more useful form for solid propellant work as

$$F = \dot{m}_{ex} v_{ex} = (r A_b \rho_p) v_{ex} \quad (\text{Eq. 14})$$

This equation is the key to solid rocket design; v_{ex} is more or less constant because it depends on the slowly-changing I_{sp} , and ρ_p is constant, but we are free to vary A_b and r at will. Actually, we find that for any solid propellant, once a nozzle throat size has been chosen, the choice of burning area determines P_c and thus burning rate, with the following being specific characteristics of black powder:

$$P_c = 0.1583 (A_b/A_t)^{1.124} \quad (\text{Eq. 15})$$

$$r = 0.0087 e^{0.00185(T_i - 295)} P_c^{0.462} \quad (P_c \leq 5 \text{ atm}) \quad (\text{Eq. 16})$$

$$r = 0.0148 e^{0.00185(T_i - 295)} P_c^{0.133} \quad (P_c \geq 5 \text{ atm}) \quad (\text{Eq. 17})$$

This last pair of equations shows not only the variation of burning rate with pressure, but also how changing local environment can affect engine performance; T_i is usually just atmospheric temperature, and as it increases so will burning rate and thrust. Normal temperature variations have little effect on total and specific impulse, although about a 5% increase in both is noted at temperatures over 110°F (316°K). Chamber pressure, however, significantly affects burning efficiency, and a rough correlation between pressure and I_{sp} can be expressed as

$$I_{sp} = 24.5 P_c^{0.607} \quad (P_c \leq 5 \text{ atm}) \quad (\text{Eq. 18})$$

$$I_{sp} = 47.6 P_c^{0.193} \quad (P_c \geq 5 \text{ atm}) \quad (\text{Eq. 19})$$

The characteristic variation of burning rate with pressure is constant for a particular fuel and can only be changed by the addition of various chemicals; up to 10% aluminum powder is sometimes added to black powder to increase its burning rate, combustion temperature, and specific impulse. Even without this metal, black powder is an electrical conductor, particularly in the gaseous post-combustion phase.

A solid engine's burning surface area is determined by the selection of the initial geometry of the propellant grain, keeping in mind that any propellant is consumed in a direction perpendicular to the ignited and burning grain surface, at a speed equal to the burning rate. For a given casing size, the minimum burning area is the cross-sectional area of the inside of the casing, and an engine with the simple geometry giving this burning area is called an end-burner. Its burning area is constant. Because of their method of manufacture, black powder engines are limited to this grain design, to a design with one or more holes of a simple shape extending down the inside of the grain (a cored-burner) or to a design combining these features (a cored-end-burner). These last two have burning areas that vary with time, and are the most commonly used designs. The addition of a center core increases the possible burning area, and thus thrust, particularly as the core burns outward and its surface area expands. Many unusual geometry variations are possible with castable-type professional propellants, making it possible to obtain a thrust-time curve of virtually any desired shape with them. Figure 3 shows the development of burning surfaces for two typical model rocket engines, and the effects of this on their thrust. The performance (particularly thrust) of deeply-cored engines can be noticeably affected by the choice of ignition method. An igniter which lights the entire core surface at once, or which lights the deepest portion first, will give the engine a more rapid thrust buildup and higher peak thrust than will an igniter producing ignition of the grain on the end near the nozzle first.

In normal engines, the delay charge acts as a diaphragm to contain chamber pressure while the last part of the grain is consumed. Actually, a small amount of propellant will remain unburned in any engine. These propellant "slivers" represent an increasingly large percentage of the propellant mass as engine size decreases, and this plus the fact that a greater portion of the engine's life is spent at inefficient lower pressures causes a decrease in the delivered specific impulse. Booster engines, lacking a delay charge, use the upper portion of the grain as a diaphragm. This eventually becomes too thin to contain the pressure and ruptures, sending burning propellant on the casing walls. Because their gases do not go through the nozzle, these slivers do not produce useful impulse, so booster engines contain a little extra propellant to ensure that they deliver the same I_T as engines with delays. If, however, the upper engine is bonded to the booster casing as in tandem engines, these slivers are made available for impulse. They tend to burn all at once at burnthrough, producing a spike on the end of the booster thrust curve which adds approximately 0.5 Nt-sec. to the I_T of 18 and 23 mm. engines.

To bring all of this discussion together somewhat, let's list in order the steps involved in designing a black-powder model rocket engine:

- 1.) Determine desired thrust vs. time characteristics. This determines the general geometry of the grain (cored, cored-end-burning, etc.) and sets fairly narrow limits on possible burning surface areas for a given casing size.
- 2.) Select a desired chamber pressure. It should be high enough to permit efficient combustion but not so high as to risk engine rupture. Through Eqs. 16-17, this determines burning rate.
- 3.) Select a nozzle area ratio (ϵ). This determines nozzle efficiency (or C_F) and affects engine size and weight.
- 4.) Select a nozzle throat area. Through Eq. 15, this and P_c determine the exact burning area and grain geometry required.
- 5.) If Eq. 9 and Eq. 14 do not now give exactly the desired thrust, adjust P_c , A_t , or grain geometry a little and start over.

Although Eqs. 15-17 exactly express the interdependence of P_c , A_b/A_t , and γ , they are somewhat inconvenient to use, so this information is usually presented graphically on logarithmic-scale paper and called a p-Kn-r chart ($Kn = A_b/A_t$). The chart for black powder is shown in Figure 4.

The details of how model rocket engines are made and the exact chemical composition of their fuel, delay charge, etc. are kept secret by the manufacturers as proprietary information, but much general information is available and some of the rest can be deduced by studying X-rays of the engines. The engines are made by hydraulically ramming powdered nozzle clay, loose black powder, etc. into the casing.

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Black Powder Model Rocket Engines *(continued from page 10)*

Since the unrammed powder is much less dense than the final product, not all of it can be put in at once; in large (E and F) engines, up to 7 charges are used in a single grain. The pressure of the ramming causes the paper engine casing to bulge outward. The lip of this bulge, plus friction between grain (or nozzle) and casing, are all that retains the grain (and nozzle) against chamber pressure; if this pressure gets too high, if the casing was bulged less than usual during manufacture, or if cold weather has caused the grain to shrink, this restraining force is overcome and either the grain or the nozzle is shot out.

Minor variations in the loaded mass and density of both propellant and delay charge are inevitable, and are responsible for normal variations translate into a much larger percentage variation in delay performance. There does not appear to be a general correlation between engine initial mass and subsequent performance, except for extreme cases—i.e., an unusually (at least 5%) heavier than average engine will probably have a greater I_T than an unusually light one.

One of the major sources of gross performance variations is a result of the successive-ram method of manufacture. In less than one percent

of modern machine-made engines, there is a significant change in propellant density, an air gap, or a piece of casing paper peeled off by the ram, between one pair of successive propellant charges in the grain. Any of these will cause a sudden variation in burning area or burning rate, and thus in chamber pressure and thrust. If the pressure varies upward too much, a catastrophic failure will result.

A second major source of catastrophic engine failures is cracks created in the grain after manufacture, either by rough handling or by cycling between hot and cold temperatures. These cracks act as additional burning surfaces, causing abrupt increases in chamber pressure which are usually extreme enough to cause blowthroughs, and which may be so rapid that the casing is ruptured before a blowthrough can occur.

This article has summarized the non-proprietary information currently available on black powder model rocket engines, and has provided an introduction to solid fuel rocket engine ballistics. For more information on solid propulsion in general, consult one of the references listed on page 8. ▲